EO-1 Technology Transfer Report for the Carbon/Carbon Radiator

August 16, 2001

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DOCUMENT CHANGE RECORD

REVISION	DESCRIPTION	DATE	APPROVAL
-	Initial Release		

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1.0 INTRODUCTION

The Carbon-Carbon Radiator (CCR) shown in Figures 1 and 2 is a sandwich composite panel with facesheets made of carbon fibers in a carbon matrix. The EO-1 flight panel is coated with an epoxy encapsulant (Figure 1.1) to prevent particle contamination of sensitive instruments on board EO-1, and provide additional strength to the panel. The external surface (Figure 1.2) of the CCR panel is coated with silver Teflon as required by the EO-1 spacecraft thermal design.

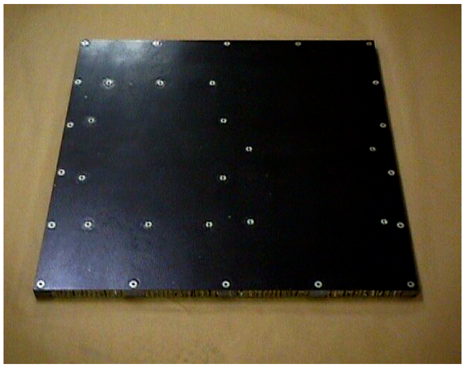


Figure 1.1
Carbon-Carbon Panel Internal Surface

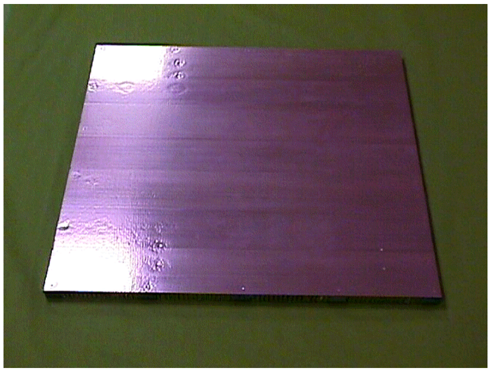


Figure 1.2
Carbon-Carbon Panel External Surface

The panel was built by the Carbon-Carbon Spacecraft Radiator Partnership (CSRP). The CSRP is an informal partnership established to promote the use of carbon-carbon on spacecraft.

Figure 1.3 - CSRP

The CSRP was started by Howard Maahs of NASA Langley and Elizabeth Shinn of Wright Patterson Air Force Base. The CRSP quickly grew to include an informal partnership with members from government and industry that include: *NASA Langley, NASA/Goddard, Air Force at Wright Patterson and Phillips Lab, Naval Surface Warfare Center, TRW, Lockheed Martin, Amoco, B.F. Goodrich, Materials Research & Design, Swales Aerospace.*

2.0 TECHNOLOGY DESCRIPTION

Carbon-Carbon (C-C) is a special class of composite materials in which both the reinforcing fibers and matrix materials are made of pure carbon. The use of high conductivity fibers in C-C fabrication yields composite materials that have high stiffness and high thermal conductivity. The primary thermal function of the EO-1 CCR is to radiate the 27.8 watts generated by the EO-1 Power Supply Electronics (PSE) and the 16.3 watts (peak power) generated by Linear Etalon Imaging Spectral Array/Atmospheric Corrector (LEISA/AC) electronics boxes. The panel is also a structural member and must support the combined weight of the PSE (50-lb) and the LEISA/AC (10-lb) boxes and the dynamic and static loads during EO-1 integration, launch and orbit induced stresses.

The CCR is a 28.62” x 28.25” composite panel with two .022” thick Carbon-Carbon facesheets bonded to a 1” 5056 aluminum honeycomb core with a density of 2.1 lb/ft^3 weighing approximately 5.5 lb (Figure 2.0).

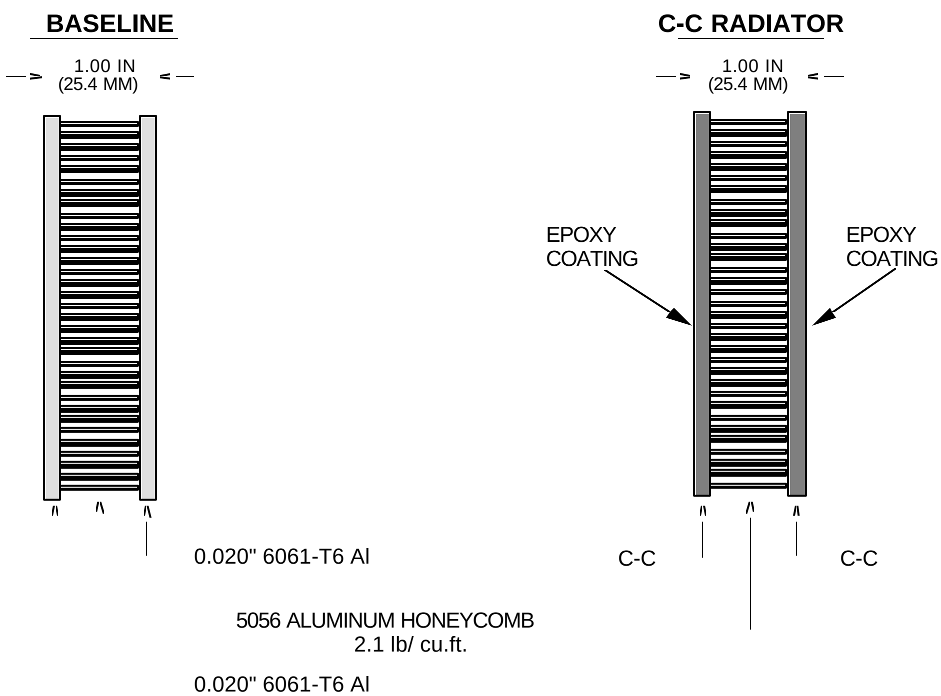


Figure 2.0

The panel is attached to the EO-1 spacecraft bus through 18 attachment points at the perimeter and supports the two electronics boxes through 14 attachment points on the interior with 8 additional inserts to support EO-1 GSE. The CSRP selected the C-C facesheet design based on a material trade study by Materials Research and Design. Each facesheet is comprised of two plies of 5-harness stain weave fabric as shown in Figure 2.1. The fabric is constructed from P30X carbon fibers and the carbon matrix is introduced by Chemical Vapor Infiltration (CVI). B.F. Goodrich fabricated the facesheets and Lockheed Martin designed the perimeter and interior attachment point configurations.

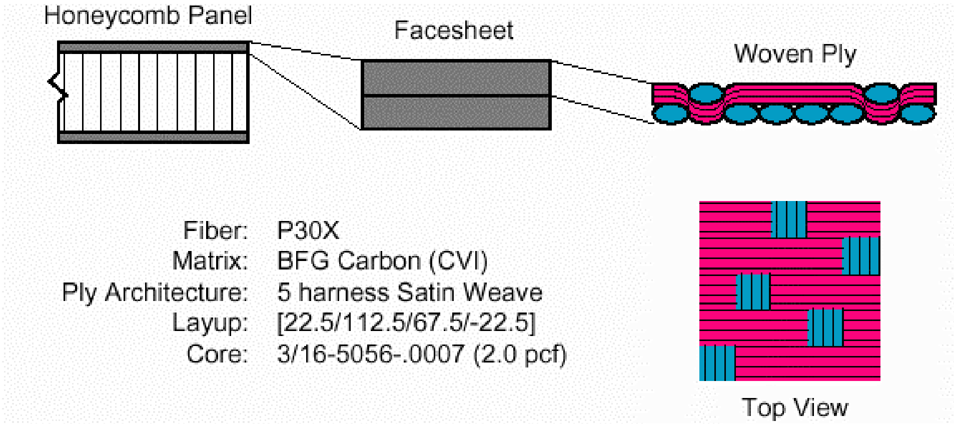


Figure 2.1

Clearance through-hole diameters in the perimeter inserts were also specified by Swales Aerospace to reduce loads induced the by thermal mismatch between the aluminum spacecraft and GC panel. An illustration of the insert design is shown in Figure 2.2.

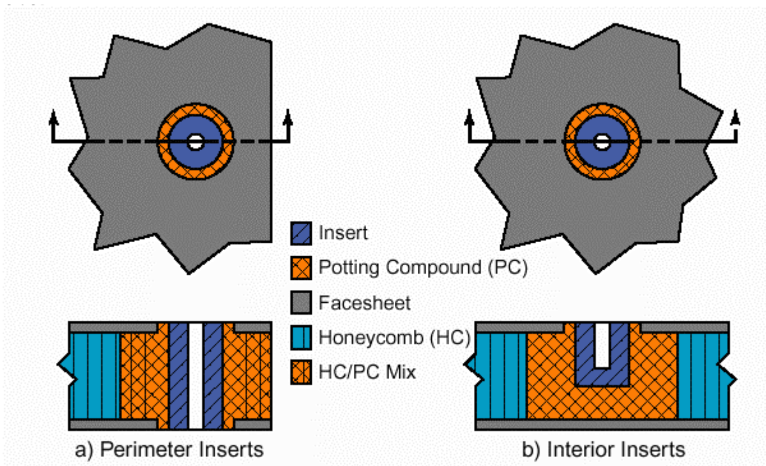


Figure 2.2

3.0 TECHNOLOGY VALIDATION

The Carbon-Carbon Radiator panel was initially instrumented with six (6) thermistors on the internal facesheet and one (1) thermistor on the external (space viewing) facesheet. However, as part of the plan to accommodate the EO-1 calorimeters, one of the thermistors was removed from CCR. The figure below shows the location of the 6 thermistors and the one that was removed.

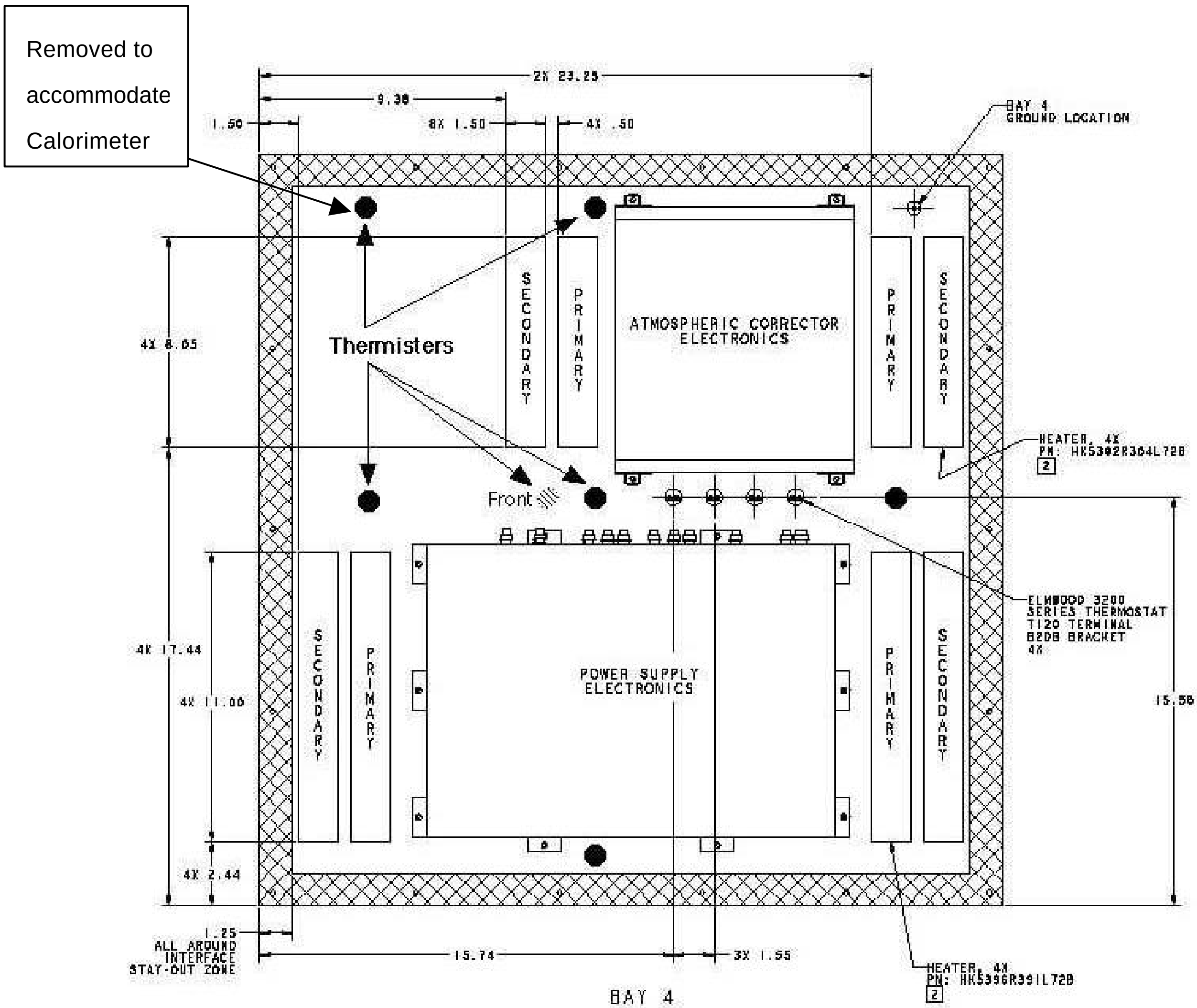


Figure 3.0.1

A simplified geometric math model (GMM) and thermal math model (TMM) were developed from the detailed EO-1 spacecraft model for this analysis effort. The GMM's are Thermal Synthesis System (TSS) models (Figure 3.0.2) and the TMM is a Systems Improved Numerical Differencing Analyzer (SINDA) model. The EO-1 early on-orbit spacecraft model correlation was excellent providing a good basis for this correlation effort.

Bay 4: Power Supply Electronics			
PSE Box Dissipation	DCE	Standby	Safe
	27.8	27.8	27.0
LEISA/AC Box Dissipation	16.3	0.0	0.0
Heater Resistance	75.7 ohms		
Heater Power	21V	28V	35V
	5.8	10.4	16.2
External Radiator Size	= 1408.59 sq.cm. = 194 sq.in. (12.0 x 16.25)		
Thermal Design Description	Box is black anodize, chotherm at box/panel interface, AgTe radiator		

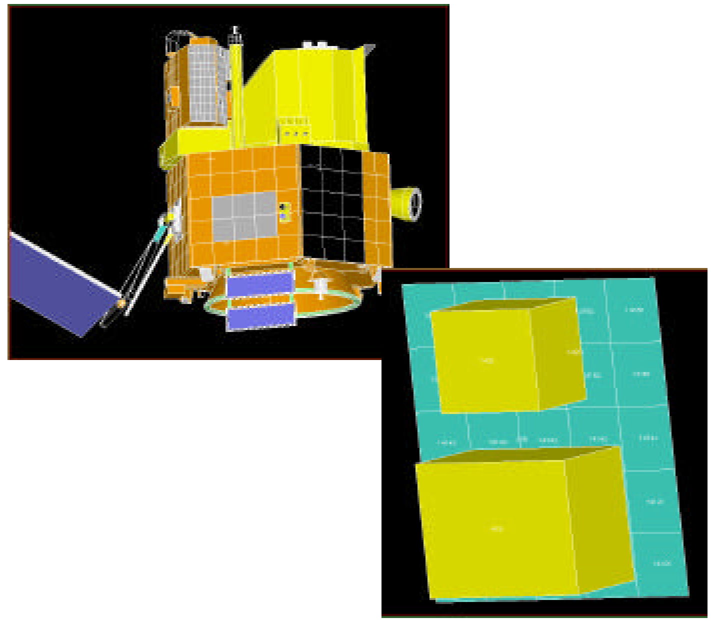


Figure 3.0.2

3.1 GROUND TEST VERIFICATION

Prior to spacecraft level testing the Carbon-Carbon radiator panel was subjected to four thermal vacuum cycles each consisting of a hot soak at 60°C for four hours and a cold soak at –20°C for four hours (Figure 3.1.1). A thermal balance test was performed at the completion of the thermal vacuum cycle test.

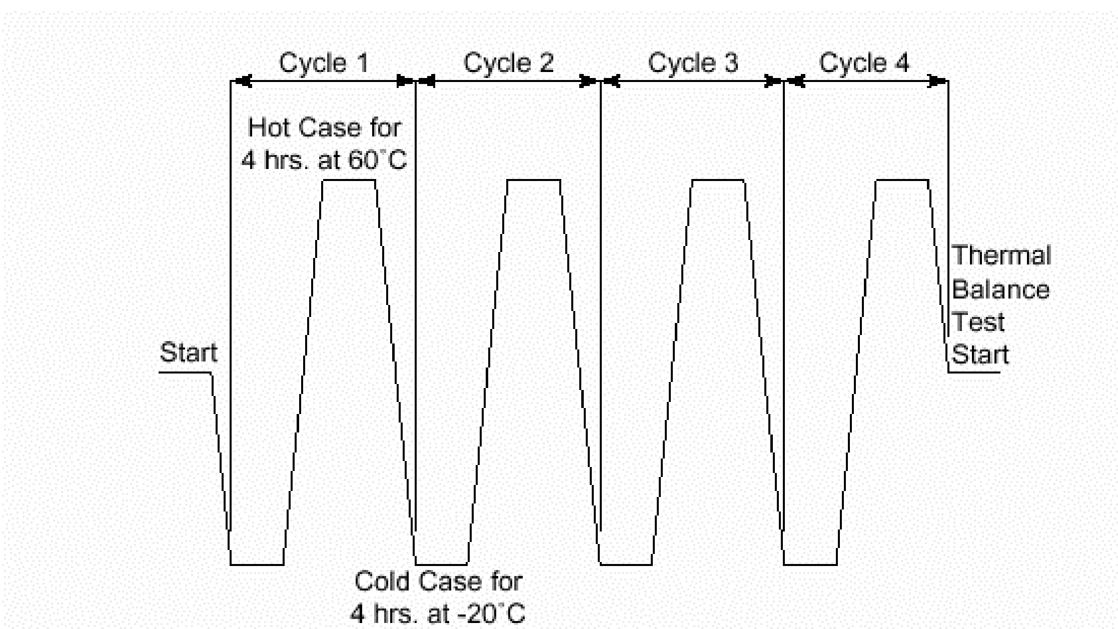


Figure 3.1.1

Figure 3.1.2 shows an illustration of the CCR component level thermal vacuum/balance test setup.

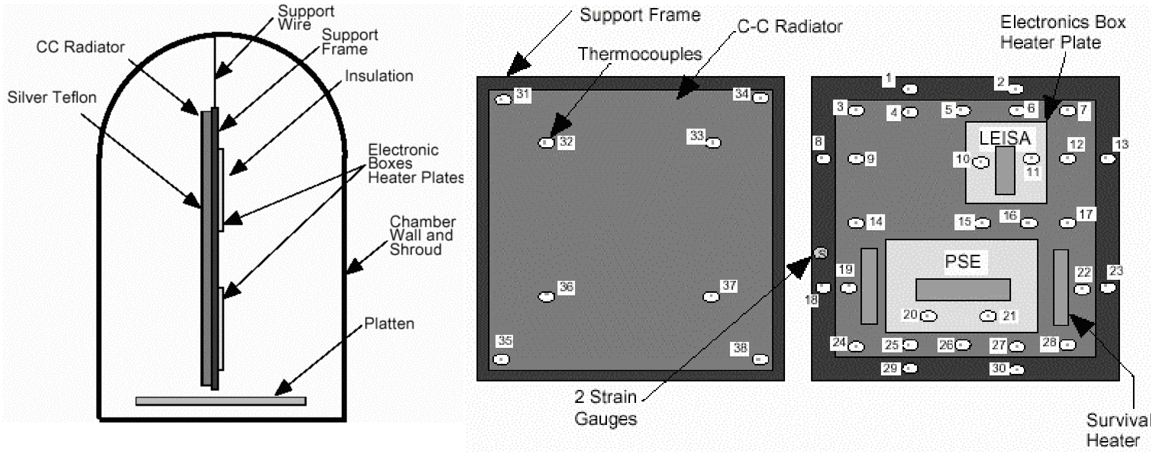


Figure 3.1.2

The results of the component level thermal balance test are shown in Table 3.1. The cold case assumed 10 watts for the PSE and 0 watts for LEISA/AC and the hot case assumed 50 watts for the PSE and 30 watts for the LEISA/AC. The test results compared very well with the model analysis.

Table 3.1

Location	Cold Case Actual	Cold Case Analysis Model	Hot Case Actual	Hot Case Analysis Model
TC15	-9°C	-11°C	28°C	30°C
TC36	-11°C	-13°C	18°C	18°C

In addition to the component level thermal vacuum/balance testing, the CCR also went through two spacecraft level thermal vacuum tests. The first test included a comprehensive thermal balance test. Both tests had the CCR panel in the flight configuration.

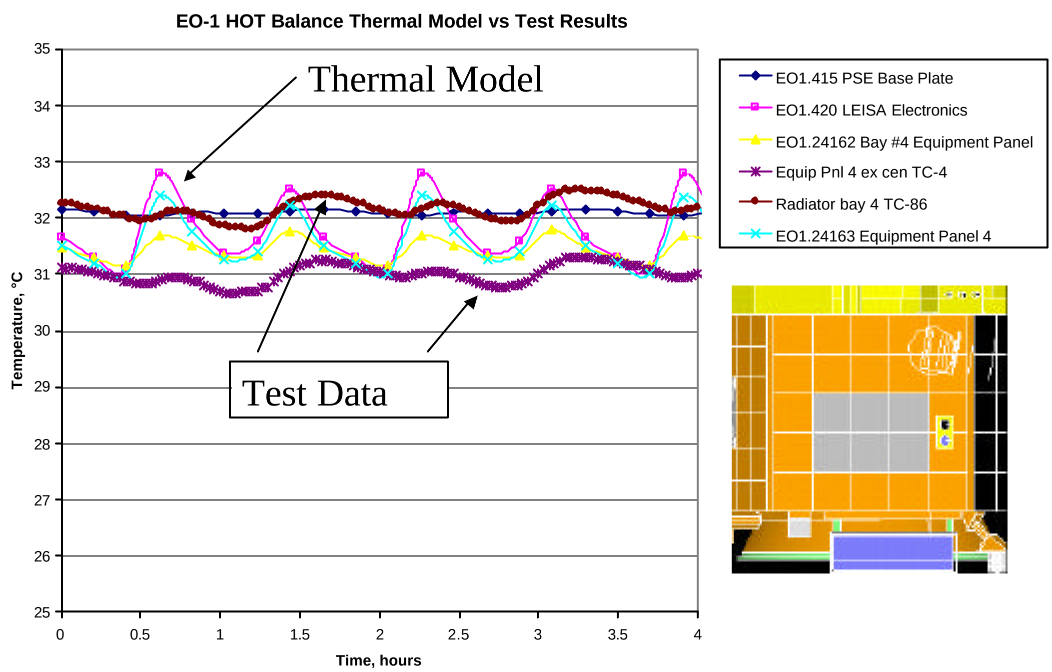
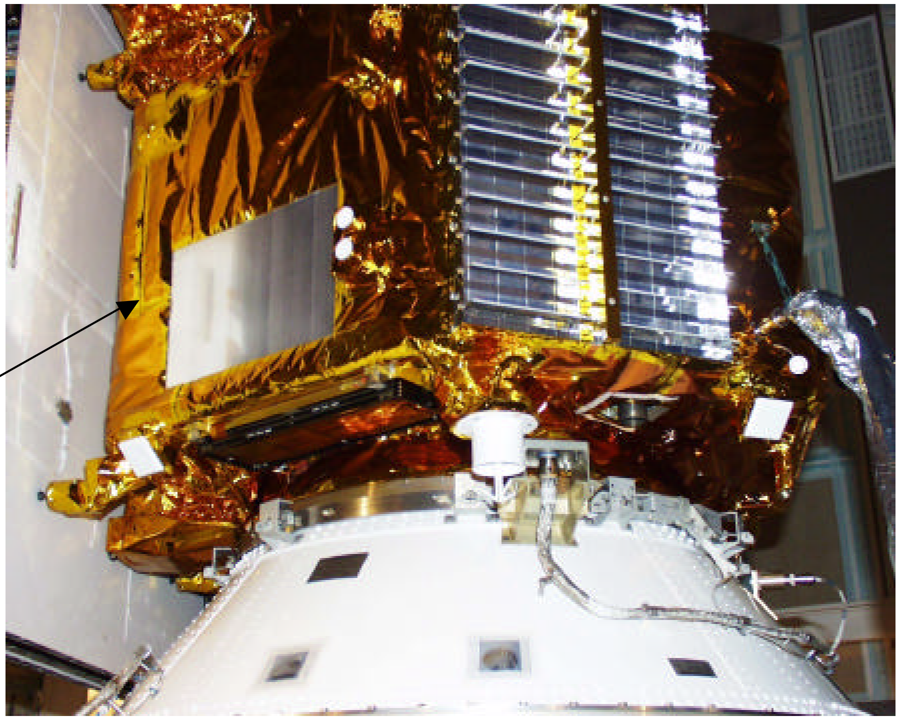


Figure 3.1.3

3.2 ON-ORBIT TEST VALIDATION

The panel gradients predicted by the model correlated within tenths of a degree of the gradients observed in the flight data.

The EO-1 image displayed at the right shows the CCR panel in its final configuration at the launch pad. The Silver Teflon radiator area is 12” x 16.25”. The remaining external surface is covered with MLI having a 3 mil Kapton outer layer.



The on-orbit thermal conductivity correlation for the CCR panel resulted in thermal conductivities close to the reported value of 230 W/m-K. The k(horizontal) is 295W/m-K, and the k(vertical) is 208 W/m-K. I used these values in the model. It appears that the model correlation indicates that the carbon-carbon layout is not isotropic as reported, and that the composite layout appears to have anisotropic properties. Having only five temperature sensors on the panel surface provides additional uncertainty in the thermal conductivity correlation. Additional sensors would provide a better understanding and mapping of the panel temperatures.

The pre-flight thermal conductivity value for the honeycomb core was based on empirical data for composite panels with aluminum facesheets. The decrease in thermal conductivity (approx. 56%) for the honeycomb core for the CCR may be attributed to the specific manufacturing of this panel.

The values shown in Table 4.1.1 were used in the SINDA model:

Table 4.1.1

	Pre-Flight / Experimental	**Flight Analysis**
K (horizontal)	230 W/m-K	295 W/m-K
K (vertical)	230 W/m-K	208 W/m-K
K (z direction)	30 W/m-K	30 W/m-K
K (honeycomb)	5245.95 W/cm-K	2936.24 W/m-K

Figure 4.1.2 shows the CCR panel flight thermistor locations and telemetry mneumonics. Thermistor, TBAY4T is the only temperature sensor on the external side of the CCR panel. All other thermistors were internal to the EO-1 spacecraft.

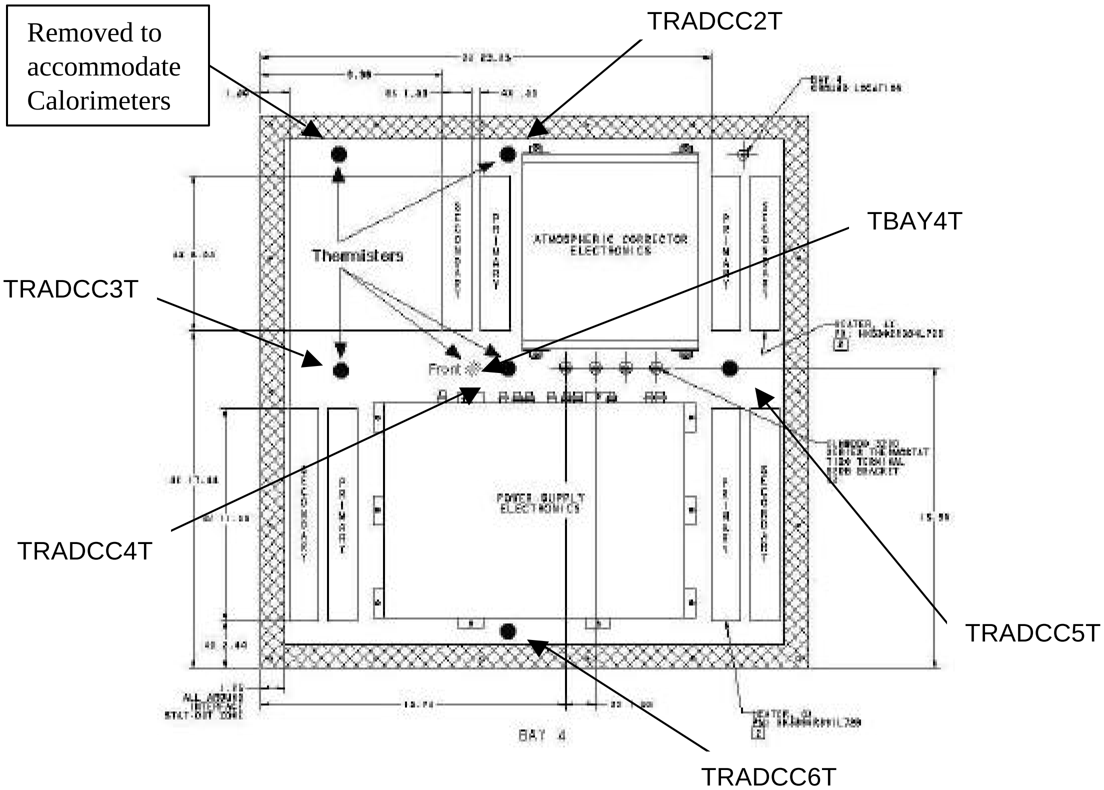
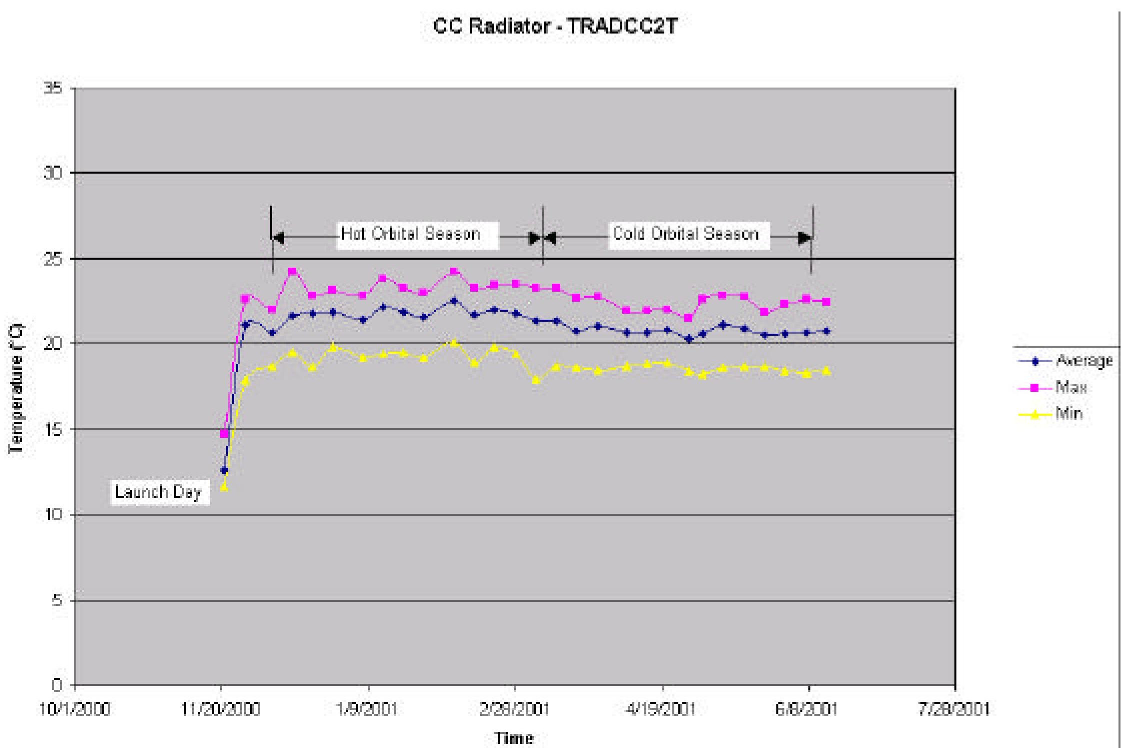
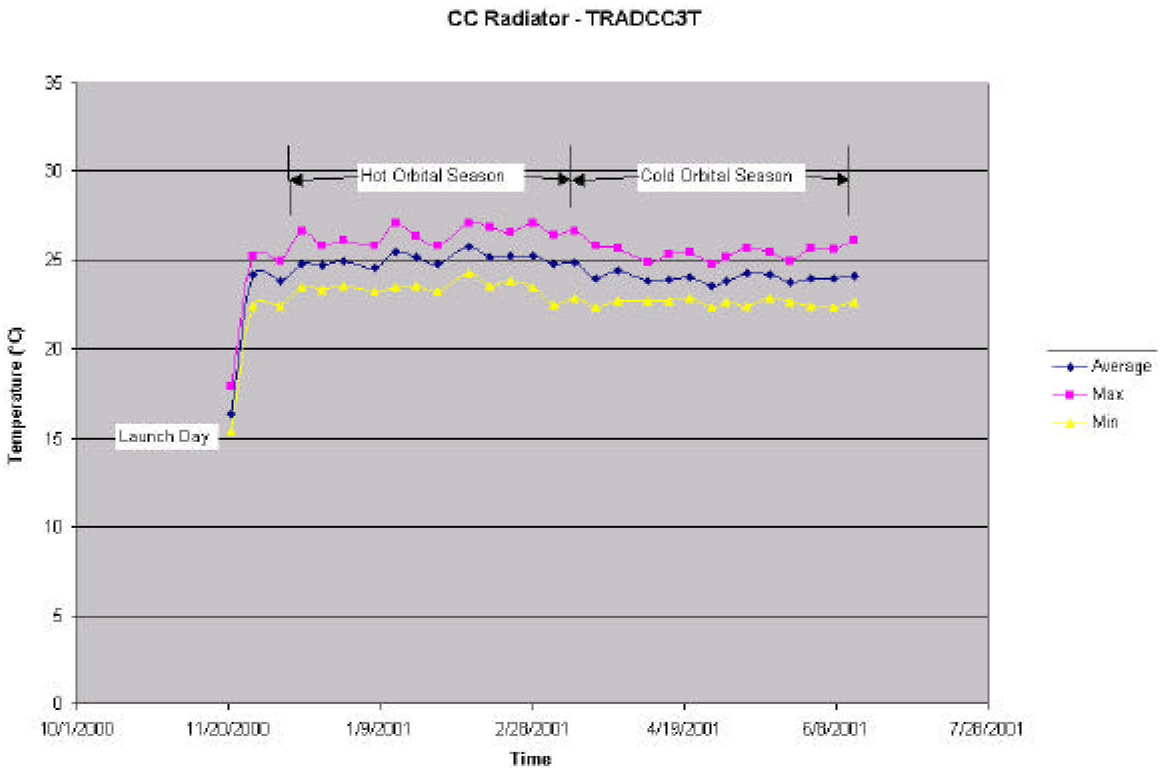
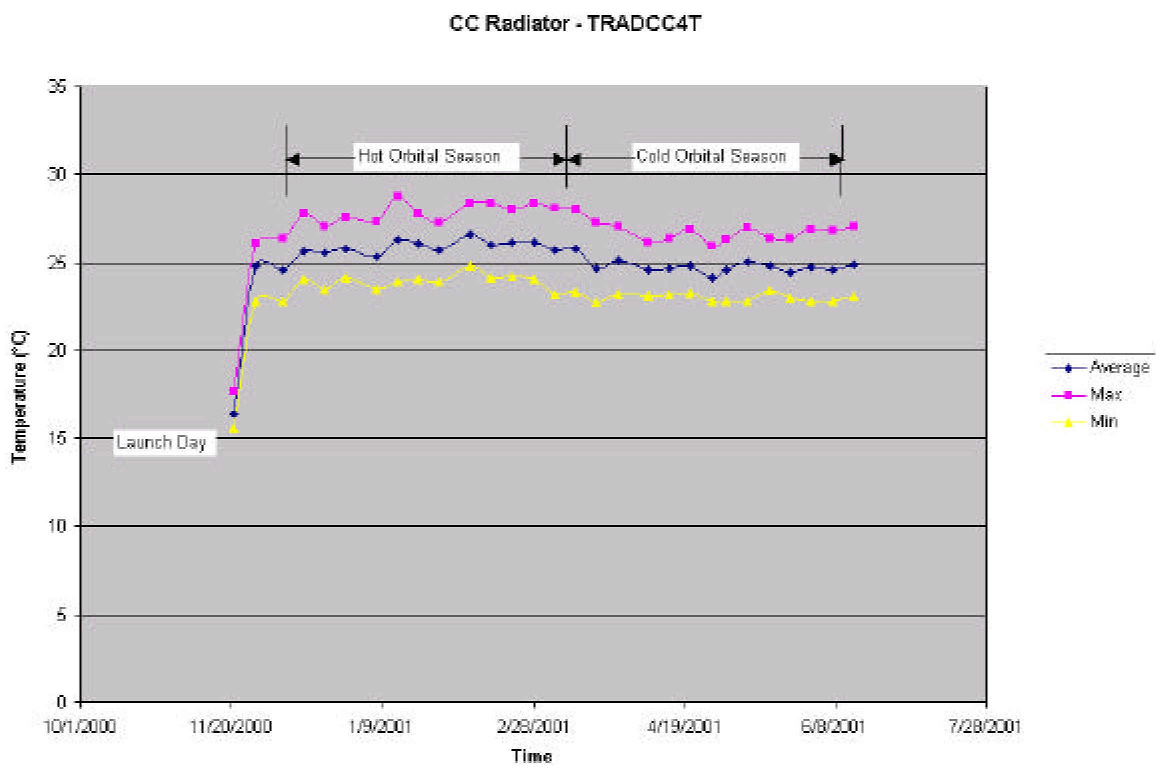

Removed to accommodate Calorimeters
TRADCC2T
TBAY4T
TRADCC3T
Thermistors
Front
ATMOSPHERIC CORRECTOR ELECTRONICS
POWER SUPPLY ELECTRONICS
SECONDARY
PRIMARY
TRADCC4T
TRADCC5T
TRADCC6T
BAY 4

Figure 4.1.2


CC Radiator - TRADCC2T
Hot Orbital Season
Cold Orbital Season
Launch Day
Temperature (°C)
Time
10/1/2000
11/20/2000
1/9/2001
2/28/2001
4/19/2001
6/8/2001
7/28/2001
Average
Max
Min


CC Radiator - TRADCC3T
Hot Orbital Season
Cold Orbital Season
Launch Day
Temperature (°C)
Time
Average
Max
Min
0
5
10
15
20
25
30
35
10/1/2000
11/20/2000
1/9/2001
2/28/2001
4/19/2001
6/8/2001
7/28/2001


CC Radiator - TRADCC4T
Hot Orbital Season
Cold Orbital Season
Launch Day
Temperature (°C)
Time
Average
Max
Min
0
5
10
15
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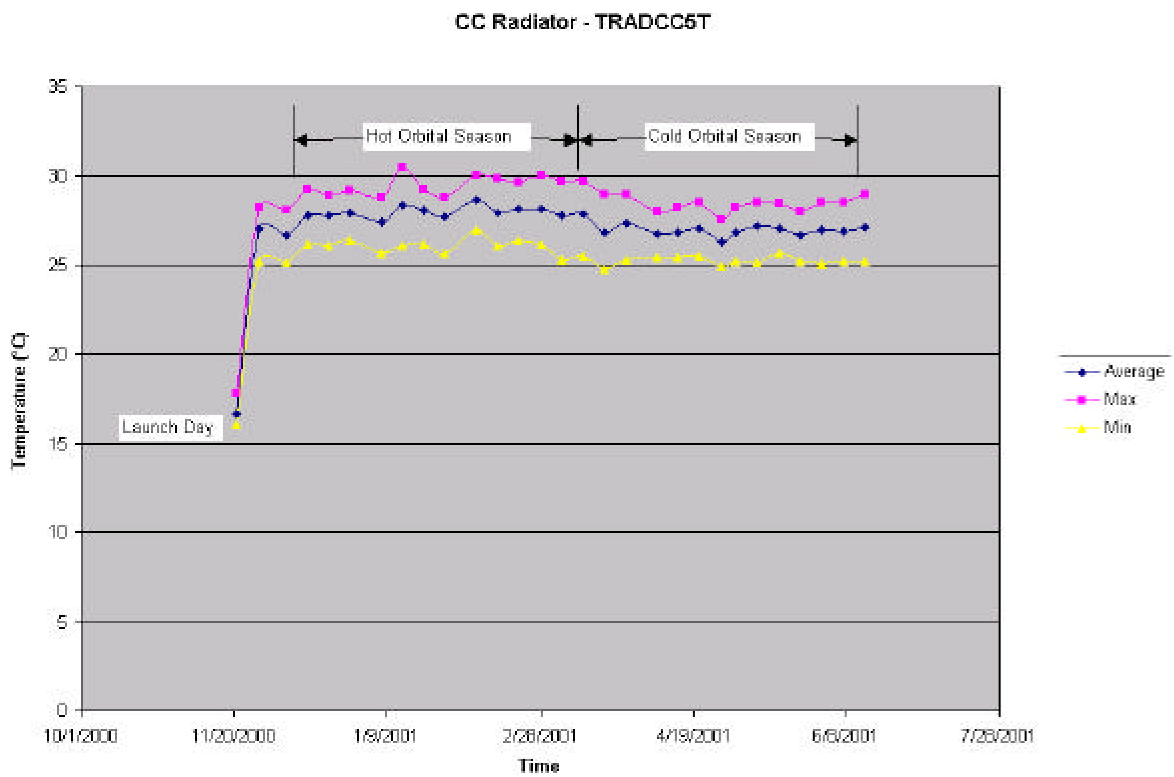
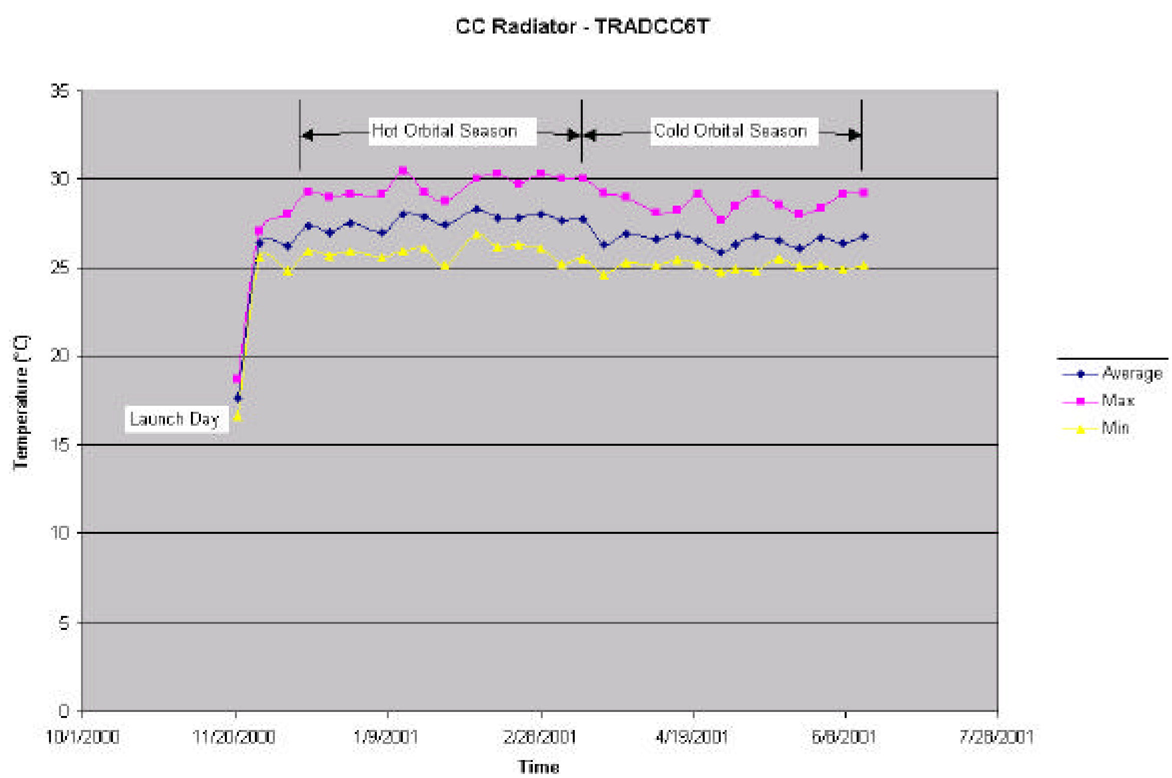
2/28/2001
4/19/2001
6/8/2001
7/28/2001


CC Radiator - TRADCC5T
Hot Orbital Season
Cold Orbital Season
Launch Day
Temperature (°C)
35
30
25
20
15
10
5
0
10/1/2000
11/20/2000
1/9/2001
2/28/2001
4/19/2001
6/8/2001
7/28/2001
Time
Average
Max
Min


CC Radiator - TRADCC6T
Hot Orbital Season
Cold Orbital Season
Launch Day
Temperature (°C)
35
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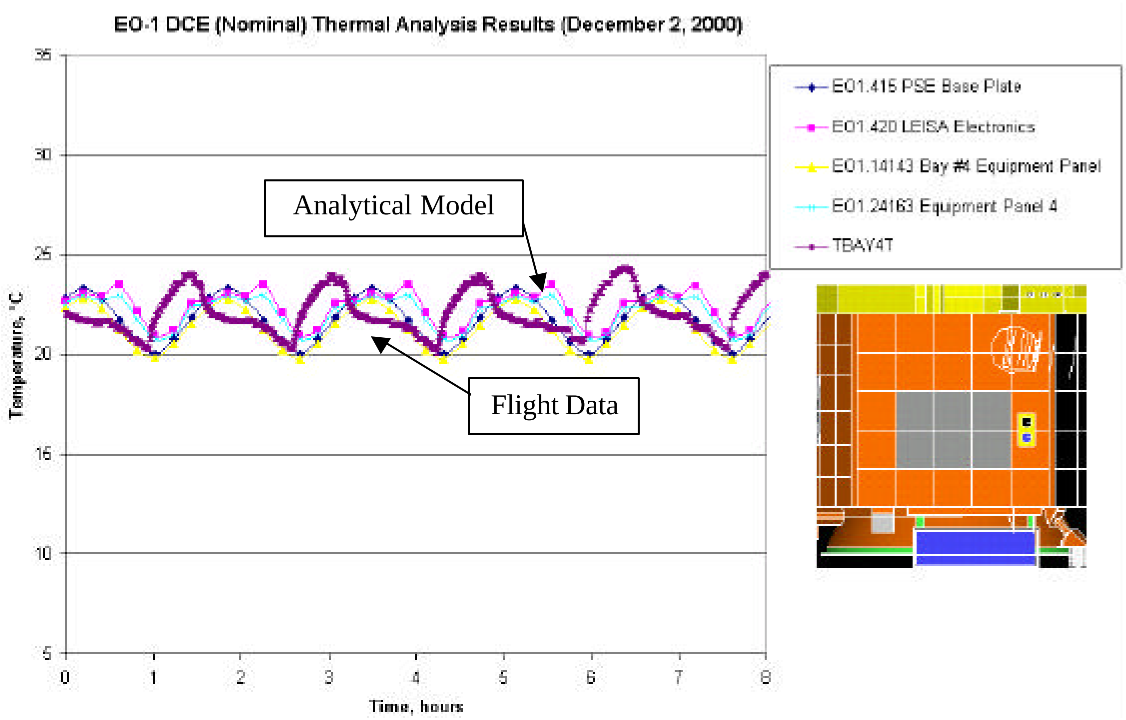
10/1/2000
11/20/2000
1/9/2001
2/28/2001
4/19/2001
6/8/2001
7/28/2001
Time
Average
Max
Min



The plot above shows how well the EO-1 thermal model predictions for the CCR panel correlated with the flight data.

3.3 ON-ORBIT USAGE EXPERIENCE

The Carbon-Carbon Radiator panel met the mission requirements and its on-orbit performance has been flawless to date.

4.0 CURRENT APPLICATIONS AND NEW APPLICATIONS POSSIBILITIES

The success of the EO-1 Carbon-Carbon Radiator Panel is complemented by some current applications and potential future applications. These include, but are not limited to:

- High Conductivity Optical Bench
- Lightweight doubler panels on low conductive composite space hardware
- Conductive bars where heat pipe applications may have been considered
- Nose cone of the space shuttle
- Re-entry vehicles that pierce the atmosphere twice, once as it enters space and on its return to earth
- Aircraft brakes
- Wing leading edges
- Engine nozzles

5.0 FUTURE MISSION INFUSION OPPORTUNITIES

Possible use of Carbon-Carbon foam as a low weight, low CTE mirror/optical bench substrate.

6.0 LESSONS LEARNED

- C-C Radiator was a success and proved that the technology can work to reduce Spacecraft weight
- C-C has a niche, especially for high temperatures
- C-C still needs further development
    - Reduction in fabrication time and cost - high conductivity “traditional” composites are more competitive
    - CTE Interface issues with heat pipes
- Redundancy a good idea - we flew the spare panel
- CSRP was a success - informal inter-agency partnership
    - Thanks to all who contributed

7.0 CONTACT INFORMATION

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8.0 SUMMARY

The thermal model results correlate very will with the EO-1 flight data. The on-orbit thermal conductivity correlation for the CCR panel resulted in thermal conductivities close to the reported value of 230 W/m-K. The k(horizontal) is 295 W/m-K, and the k(vertical) is 208 W/m-K. It appears that the model correlation indicates that the carbon-carbon layout is not isotropic as reported, and that the composite layout appears to have anistropic properties. The plan is to continue to evaluate the performance of the CCR panel for the entire 12 months mission, re-evaluate the data and provide an updated report to NASA/GSFC.

9.0 CONCLUSIONS

The CCR was a success and proved that the technology can work and should continue to find space applications, especially for high temperature thermal management applications. However, in an era when faster, better, cheaper is still a key focus for most space related projects, the C-C process still needs further development to obtain ways to reduce fabrication time and cost.

10.0 TECHNICAL REFERENCES

PROCEEDINGS of the 1999 SPIE CONFERENCE

THERMAL and MECHANICAL PERFORMANCE of a CARBON/CARBON COMPOSITE SPACECRAFT RADIATOR
Jonathan Kuhn, Steve Benner, Dan Butler and Eric Silk
NASA/Goddard Space Flight Center, Greenbelt, MD 20771

UNIQUE PARTNERSHIP YIELDS SPACECRAFT IMPROVING RADIATOR PANEL
Timothy Anderl, United States Air Force News Release
AERONAUTICAL SYSTEMS CENTER PUBLIC AFFAIRS
WRIGHT-PATTERSON AFB OH 45433